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APOLLO SERVICE PROPULSION SYSTEM INJECTOR COLD FLOW TEST

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T. L. Ridings and R. E. Southerlan

ARO, Inc.

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APOLLO SERVICE PROPULSION SYSTEM INJECTOR COLD FLOW TEST

T. L. Ridings and R. E. Southerlan
ARO, Inc.

FOREWORD

The work reported herein was done at the request of the National Aeronautics and Space Administration (NASA), Manned Spacecraft Center (MSC), Houston, Texas, under System 921E.

The results presented were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC), Arnold Air Force Station, Tennessee under Contract F40600-69-C-0001. The tests were conducted from December 15, 1967, to February 2, 1968, under ARO Project No. ST1805, and the manuscript was submitted for publication on May 28, 1968.

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This technical report has been reviewed and is approved.

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Director of Test

ABSTRACT

A full-scale production Apollo Service Propulsion System Injector was modified to accommodate detailed instrumentation and visual observation capability during a series of propellant rapid expansions to high vacuum conditions to determine the venting characteristics of the injector. It had been suspected that after short burns of the engine at altitude, evaporative freezing of the residual propellants in the injector might result in clogged passages which could prevent safe restarts for extended time periods. Test results indicate that 5 min of venting between engine firing is adequate if propellant and injector temperatures are maintained above 55°F.

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SECTION I
INTRODUCTION

Current Apollo missions and abort maneuvers require the Service Propulsion System (SPS) to perform short burns with engine restart, or a series of short burns with varying coast periods between burns. The burn period could be as short as 0.5 sec, with coast periods from 5 sec to several hours. The short burn results in little heat being added to the system. Since heat is required to vaporize any residual propellant that remains in the injector manifolds after engine shutdown, propellant may be evaporatively frozen and remain in the injector for long periods of time, making predictable engine restarts doubtful. Frozen propellant in the manifold could restrict flow during attempted restarts to significantly affect engine performance.

The oxidizer, nitrogen tetroxide (N_2O_4), because of its higher vapor pressure, should vent from the manifolds more rapidly than the fuel, Aerozine-50 (AZ-50). During short coast periods, it is possible that the oxidizer manifold will vent properly while the fuel manifold remains partially primed for some additional period of time with frozen fuel. This would provide, in effect, a fuel lead, and an attempted engine restart under these conditions could very likely result in engine damage.

This test series was planned to provide data concerning the SPS injector manifold venting characteristics, the evaporative freezing effects, and the sublimation rate of any propellant remaining in the injector manifolding.

This report covers the cold flow testing of a modified full-scale Apollo SPS injector. The modifications were for instrumentation and photographic coverage during tests.

One of the basic requirements of this test was that the SPS injector be initially filled with the particular fluid or fluids and be confined in a chamber at the vapor pressure of the fluid(s) concerned. When the desired initial temperatures were attained, it was specified that this confined chamber pressure be rapidly reduced to a vacuum level sufficient to provide the evaporative conditions that would be experienced in space.

The test objectives were to: (1) determine the propellant venting characteristics of the SPS injector under vacuum conditions, (2) determine the relationship between the injector venting characteristics and the initial propellant and hardware temperature, (3) determine if propellant freezing occurs in the injector manifolding and the mass of frozen propellant remaining, (4) determine the sublimation rate of the frozen propellant, and (5) determine if frozen propellant tends to plug injector orifices and flow passages.

SECTION II
APPARATUS

2.1 TEST ARTICLE

2.1.1 General Description

The Apollo Service Propulsion System is shown in Fig. 1 (Appendix I). The modified injector is shown in Figs. 2, 3, and 4, with the instrumentation attachment points and areas for view port installations indicated.

The Apollo SPS aluminum alloy injector was designed and built by the Aerojet-General Corporation. The injector receives the propellants, oxidizer (N_2O_4) and fuel (AZ-50), through individual manifolds to separate orifice arrays. After passing through these orifice arrays, the propellants are mixed together and hypergolic combustion occurs. In these tests, the injector was oriented with the orifice arrays pointing up.

2.1.2 Modifications

The modifications for temperature and pressure measurement devices and for view ports and light ports consisted of aluminum extensions and attachment flanges welded to the injector and penetrations cut to provide access to the inside of the injector. These modifications can be seen in Figs. 5 and 6.

2.1.3 Instrumentation

The instrumentation used, its ranges, and the methods used to read and record are all outlined in Table I, Appendix II. Wiring, feedthroughs, connectors, and potting compounds used were chosen to provide the greatest possible resistance to adverse reactions with the propellants. Figure 7 shows instrumentation attachments, leads, and feedthroughs.

High- and low-speed motion picture coverage was provided as shown in Fig. 8, and the data were recorded on magnetic tape using the high-speed Universal Data System of the Aerospace Environmental Facility.

2.2 TEST CHAMBERS

2.2.1 ARC (8V)

The basic requirement of this test program, rapid evacuation of the SPS Injector, was accomplished by installing the injector in a relatively small antechamber (A/C) and attaching it by means of a large duct and valve to either of several large-volume cryogenic pumping chambers available in the Aerospace Environment Facility.

As shown in Fig. 9 the injector was mounted inside the A/C adjacent to the pumping or test support chamber. A 10-in. duct with an air-operated 10-in. valve connected the A/C to the pumping chamber.

Phase I of the Apollo SPS injector cold flow test using the oxidizer, N_2O_4, was conducted with the support of the ARC (8V) (Aerospace Research Chamber (8V)). The ARC (8V) is a stainless steel vacuum chamber 10 ft in diameter and 20 ft in length containing 600 ft^2 of liquid-nitrogen (LN_2)-cooled cryosurface and 120 ft^2 of gaseous-helium (GHe)-cooled cryoarray. This chamber is capable of holding a vacuum of 10^{-2} torr through the test.

This chamber provided vacuum pumping through the 10-in.-diam connecting valve. During the oxidizer tests, the liquid oxidizer supply tank was confined within the test support chamber to provide maximum safety conditions in the test area. The complete test support system is diagrammed in Fig. 9.

A toxic vent system was provided for use during chamber repressurization cycles so that the vapors of N_2O_4 could be safely exhausted outside the chamber laboratory building.

2.2.2 ARC (7V)

Phases II and III of the Apollo SPS injector cold flow test, using the fuel, Aerozine-50 (in Phase II), and the fuel plus Freon MF®, a "simulated" oxidizer (in Phase III), were conducted with the support of the ARC (7V) (Aerospace Research Chamber (7V)).

The ARC (7V) is a stainless steel vacuum chamber 7 ft in diameter and 12 ft in length. In Phase II, 64 ft^2 of the tube-in-sheet paneling served as the LN_2 cryosystem but later, in Phase III, the 480-ft^2 LN_2 chamber liner was required to carry the additional load imposed by the Freon MF in addition to the Aerozine-50. Figure 10 shows an artist's concept of the test installation.

A toxic vent system was also used during these tests to remove the Aerozine-50 vapors. A water-flush Aerozine-50 removal system was provided to accelerate the posttest operations procedures. Figure 11 shows a schematic of the test support system used in Phases II and III.

SECTION III
PROCEDURE

3.1 TEST SEQUENCE

The pretest preparation was the same in all phases. In the first run of each phase the main chamber and antechamber were evacuated below 10^{-4} torr using the mechanical and cryogenic pumping systems. After obtaining this pressure, the mechanical pumping system was isolated and the chamber pressure was maintained by the test chamber cryosystem. The 10-in. antechamber valve was then closed and propellant admitted to the injector. Amounts of propellant were chosen that would completely fill the supply ducts and fill the manifolds to the level at which liquid started through the center orifices of the injector. This amounted to approximately 2 lb of fuel and 3.5 lb of oxidizer. Initial temperatures were obtained by evaporating propellant from the injector when cooling was required, and by adding heated propellant when injector warming was necessary. Because of its geometry, the injector was never completely filled. Various initial temperatures were set as shown in Table II. The testing sequence involved manual and automatically controlled events as follows:

1. T-10 sec - countdown started
2. T-0 sec - data coverage started and the following automatic sequence began:
   a. T+3 sec - camera lights on
   b. T+4 sec - camera coverage started
   c. T+5 sec - 10-in. valve opens

3.2 PHOTOGRAPHIC/DATA COVERAGE

Photographic and instrumentation coverage varied in acquisition rate depending on the propellant used and the initial temperature. High-speed data coverage was used the first 60 to 120 sec, followed by periodic sampling of the data every 5 sec. Continuous photographic coverage was obtained for the first 12 to 300 sec depending on how rapidly the injector vented. A test run was terminated when propellant ice was no longer present in the injector or when ice remained longer than practical mission limits would allow. At the conclusion of the test run, the 10-in. valve was closed and test conditions were set for another run.

SECTION IV
RESULTS AND DISCUSSION

4.1 SUMMARY OF RESULTS

A total of 27 cold flow test runs were made, nine of which are plotted in Figs. 12 through 20. These figures show temperature and pressure versus time data at selected locations during the test runs. Tables II and III summarize data obtained from the instrumentation and photographic coverage. Photographs of various ice formations observed in the test are shown in Figs. 21 and 22.

4.1.1 General Results

Photographic coverage in the injector ducts and manifolds indicated that injector venting adequate for a restart occurred within 5 min after exposure to vacuum conditions for initial hardware and propellant temperatures of 55°F and above. In the oxidizer tests, such venting occurred with initial temperatures as low as 45°F. Small propellant ice particles remained in and on the injector for longer periods, but the amounts involved were not significant. However, large masses of ice that formed in the injector at initial temperatures near 30°F remained for several hours.

Some temperature measurements indicated the presence of ice long after visual observations and weight measurements had indicated that the injector had cleared. Table II presents venting times as indicated by these temperature measurements; because of the location and mounting configuration of the sensing elements, these results did not always agree with photographic observations (Table III).

4.1.2 Oxidizer Tests, Phase I

The oxidizer vented clearly in the 62 and 45°F runs. In the 33°F run large formations of ice flakes were noted that would be classed above tolerable limits for a restart. The hardware temperature in the 33°F run (Fig. 14) varied from other runs in that a low temperature of 6°F was observed in 60 sec, followed by a slight increase in temperature, then decreased to 5°F, 1110 sec into the run. This could have been caused by ice formation and movement inside the injector near the hardware temperature sensor. The long time emphasizes that once ice has formed, venting is then by a slow sublimation process.

4.1.3 Fuel Tests, Phase II

The fuel vented much slower than the oxidizer. This was expected considering the properties of the two propellants. Venting was considered adequate for a restart within 5 min after vacuum exposure for the runs above 50°F, although the instrumentation indicated longer times as discussed earlier. Some ice flake buildup was observed in the manifold area in the 52°F run, but the amount was not considered to be significant. Ice slush formed in the 33°F run that, in flight, would have prevented a successful restart.

4.1.4 Combined Fuel and Simulated Oxidizer Tests, Phase III

The data presented in Tables II and III were measured and observed at the same location as in Phase II. Photographic coverage was successfully obtained in all test runs of this phase at the fuel duct port, but successful coverage of the manifolds was obtained on only two runs. Little difference was visually noted in the venting characteristics of Phases II and III.

Initial control of the various temperatures was complicated by the fuel and simulated oxidizer having different vapor pressures. Consequently, the data spread was larger in this phase than in the first two phases. Fuel and simulated oxidizer were admitted into the injector under vacuum conditions where evaporative cooling took place as equilibrium vapor pressure-temperature conditions were established. The desired temperature conditions could only be obtained by pumping off and evaporatively cooling the system and then adding more propellant at a temperature estimated to give the desired end results. Pretest temperatures were set in both the simulated oxidizer and fuel to nearly the same values. This preconditioning was further complicated in several runs by liquid evaporating from inside the injector, condensing on the outside and dripping on the bottom antechamber port, thus obscuring photographic coverage of the oxidizer and fuel manifold view ports.

4.2 TEMPERATURE AND PRESSURE MEASUREMENTS

Numerous temperature and pressure measurements were made on the injector and the antechamber. Only eight measurements are presented in the data. Pressure measurements were made at the top of the antechamber, in the oxidizer manifold, and in the fuel duct. Five temperature measurements include: (1) one each for the oxidizer and fuel manifolds and ducts, and (2), one hardware temperature.

Pressure measurements (Figs. 12 to 20) show a rapid decrease in antechamber pressure as the 10-in. valve was opened. The differences in this pressure and that measured inside the injector indicated that choked flow conditions existed across the orifices as the propellant vented. This is the same condition that would be expected under flight conditions. Pressure data presented in the figures were measured at the same fuel duct locations in Phases II and III.

Propellant temperature sensors were located in the injector manifolds and in the bottom of the supply ducts. Table II was constructed from these measurements and on the supposition that the temperature inflection point observed after reaching the lowest temperature of the test run represented the time the probe ceased to sense cooling from evaporation liquid or sublimation, indicating that liquid or ice was no longer present in these areas. Comparing Tables II and III, the sensors indicated that propellant remained in the manifolds and ducts for longer periods of time than indicated by the photographic coverage. A probable explanation lies in the mounting configuration. The sensors were mounted in vertical wells which tended to collect and retain frozen propellant, thus making this measurement a conservative indicator of the venting time.

Position 17 was selected as the hardware temperature in the oxidizer tests (Phase I), and position 4 was chosen in Phases II and III. Large temperature gradients existed over the injector, as evaporating and subliming of the propellants occurred in a relatively small area. The sensor positions chosen were representative of the approximate average temperature of the center area of the injector for a majority of test runs.

4.3 DESCRIPTION OF THE VENTING PROCESS

Visual and photographic coverage indicated several modes of venting. Immediately after the 10-in. valve was opened, expansion of gases in solution in the propellant and the initial boiling forced a large percentage of the propellant through the orifices in liquid form. At higher temperatures (from 55 to 80°F) it is estimated that from 80 to 95 percent of the initial propellant fill vented in this manner. These estimates were arrived at by noting the propellant level in the duct view port.

This venting mode lasted less than 5 sec and was characterized by a shower of porous ice crystals as evaporative freezing took place outside the injector, Fig. 21. Most of this ice was blown clear of the injector face as it formed. In Phase II and Phase III runs, some of the liquid remained on the injector face and formed a dense ice sheet similar in appearance to water frozen at atmospheric conditions. After forming, the ice crystals, or sheets, remained up to several hours.

The venting process was observed internally through glass view ports located in the manifold and duct areas shown in Figs. 5 and 6. As the antechamber was opened to vacuum conditions, violent boiling, Fig. 22, could be seen in both the manifold and the duct. The manifold view ports cleared well in the 67 to 80°F initial temperature runs. Various sizes of ice flakes formed at 55°F, and complete visual blockage of the port occurred near 30°F. Similar action could be seen in more detail in the duct view port, Fig. 22.

As soon as the boiling had subsided enough for observation of the duct interior, the liquid level was near or below the bottom of the view port. The boiling continued, splashing liquid on the port. In general, as the internal pressure in the injector dropped, this liquid would start abruptly to freeze on the window and later to sublime. This action was most apparent in the test runs made near 50°F and occurred from 3 to 5 sec into the oxidizer runs and from 30 to 50 sec into the fuel and combined fuel and simulated oxidizer runs. Ice activity usually subsided shortly after this occurred.

Flakes and small particles of ice varying in thickness and size formed in all runs, e.g., Fig. 22. It was found, however that these ice forms would melt almost instantaneously as propellant was admitted into the injector. After observing this melting, test experience would indicate that much more of this type of ice buildup could be tolerated for restart than occurred in test runs of 55°F and above.

This was not the case at temperatures near 30°F. Within a few seconds after the boiling started, an ice slush began to form. The duct view port quickly became covered from the slush forming in place and also from large pieces falling from locations above the port. The manifold view port also became blocked, indicating that the slush formed throughout the injector. Several minutes were required after the addition of ambient temperature propellant, particularly in the fuel and fuel-simulated oxidizer runs, to melt this ice formation. Under these conditions propellant flow would be impeded if an engine restart were attempted.

4.4 SUBLIMATION RATE MEASUREMENTS

An attempt was made to determine the sublimation rate of the propellants by continuously monitoring the injector weight. This measurement was not successful primarily because of unpredictable drift in the measuring equipment that obscured the slow sublimation rates.

4.5 PREVENTION OF ICE FORMATION

The key factor in assuring successful repetitive engine starts with short venting times between burns is the prevention of ice formation. After ice is formed very little heat necessary for sublimation is transferred by the various heat-transfer mechanisms. Weightless conditions could further minimize some of these mechanisms. Various types of heating devices could be added to the injector to maintain the unit above desired minimum temperatures, but it should be remembered that such heat may be more effective in the prevention of ice than in its removal.

SECTION V
CONCLUSIONS

Visual observation into the injector and of the orifice areas indicated that venting was accomplished by liquid entrainment the first few seconds after vacuum exposure, by some boiling inside the unit, and by sublimation. Three types of ice formations were observed: porous crystals, flakes of various sizes and thicknesses, and dense ice slush. Some buildup of the crystals and flakes could be tolerated, as they quickly melted upon contact with liquid propellant. Test results would indicate that 5 min of venting between engine firings is adequate if propellant and hardware temperatures are maintained above 55°F.

APPENDIXES

I. ILLUSTRATIONS

II. TABLES

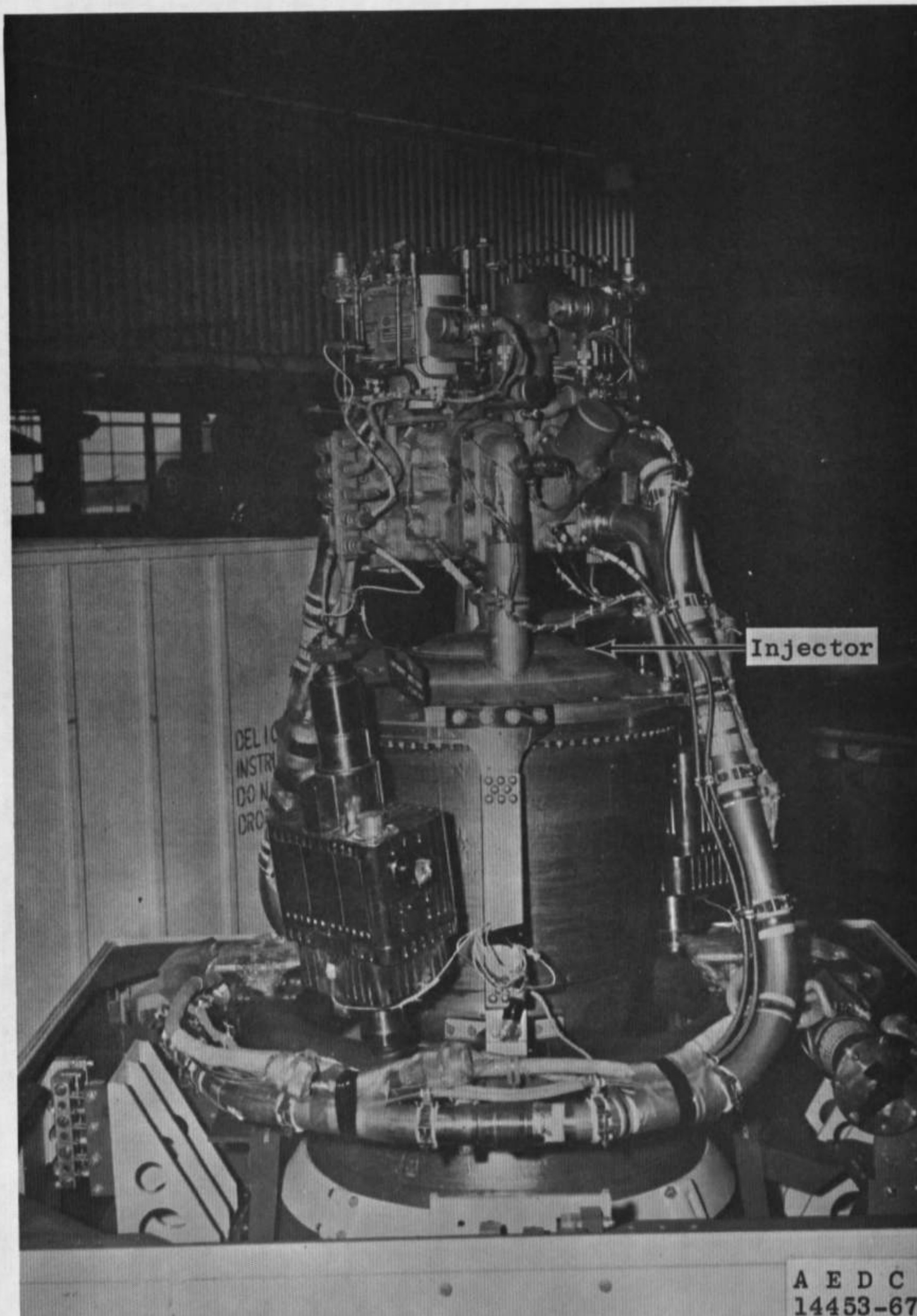


Fig. 1 Apollo Service Propulsion System (SPS)

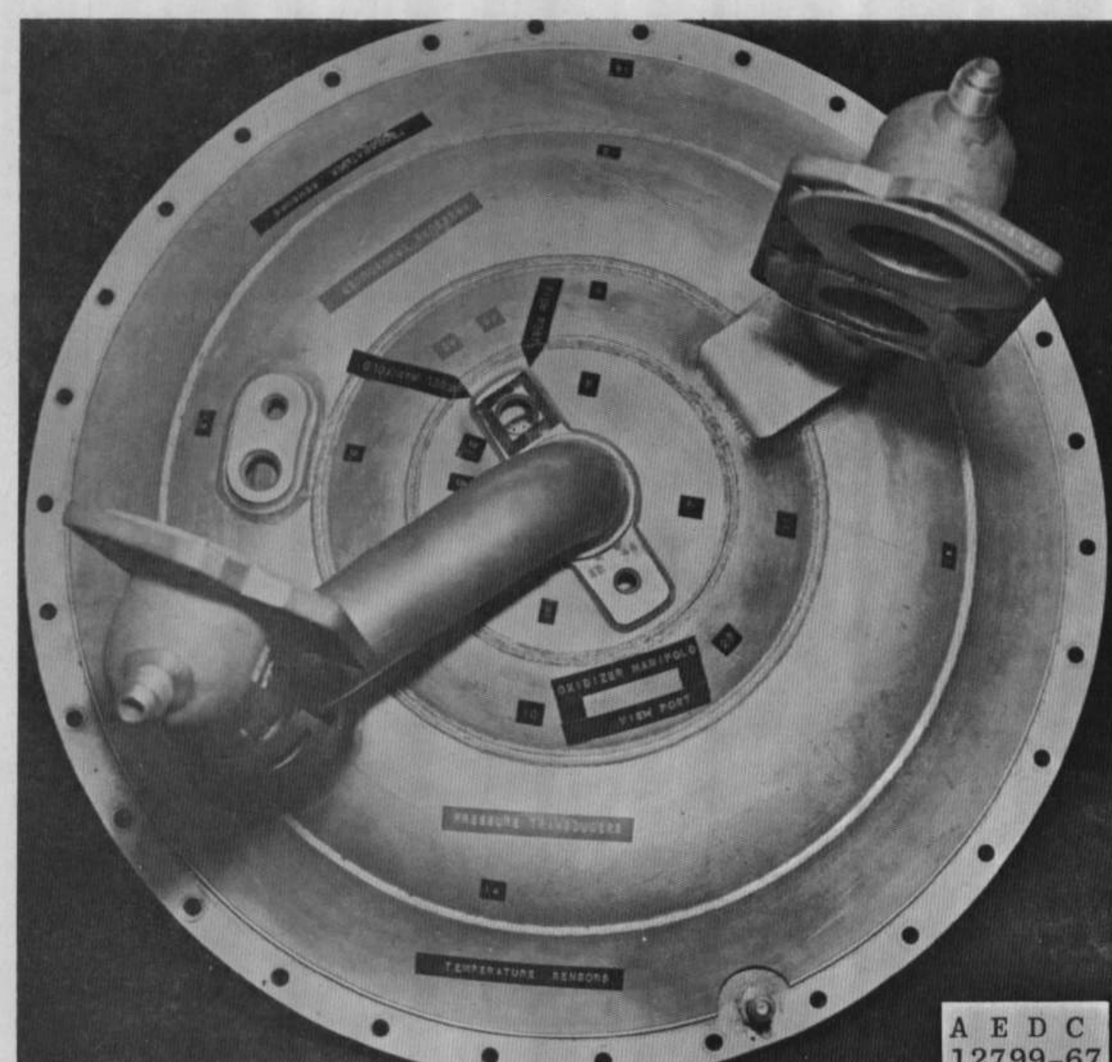


Fig. 2 Top View of SPS Injector Showing Instrumentation and View Port Locations

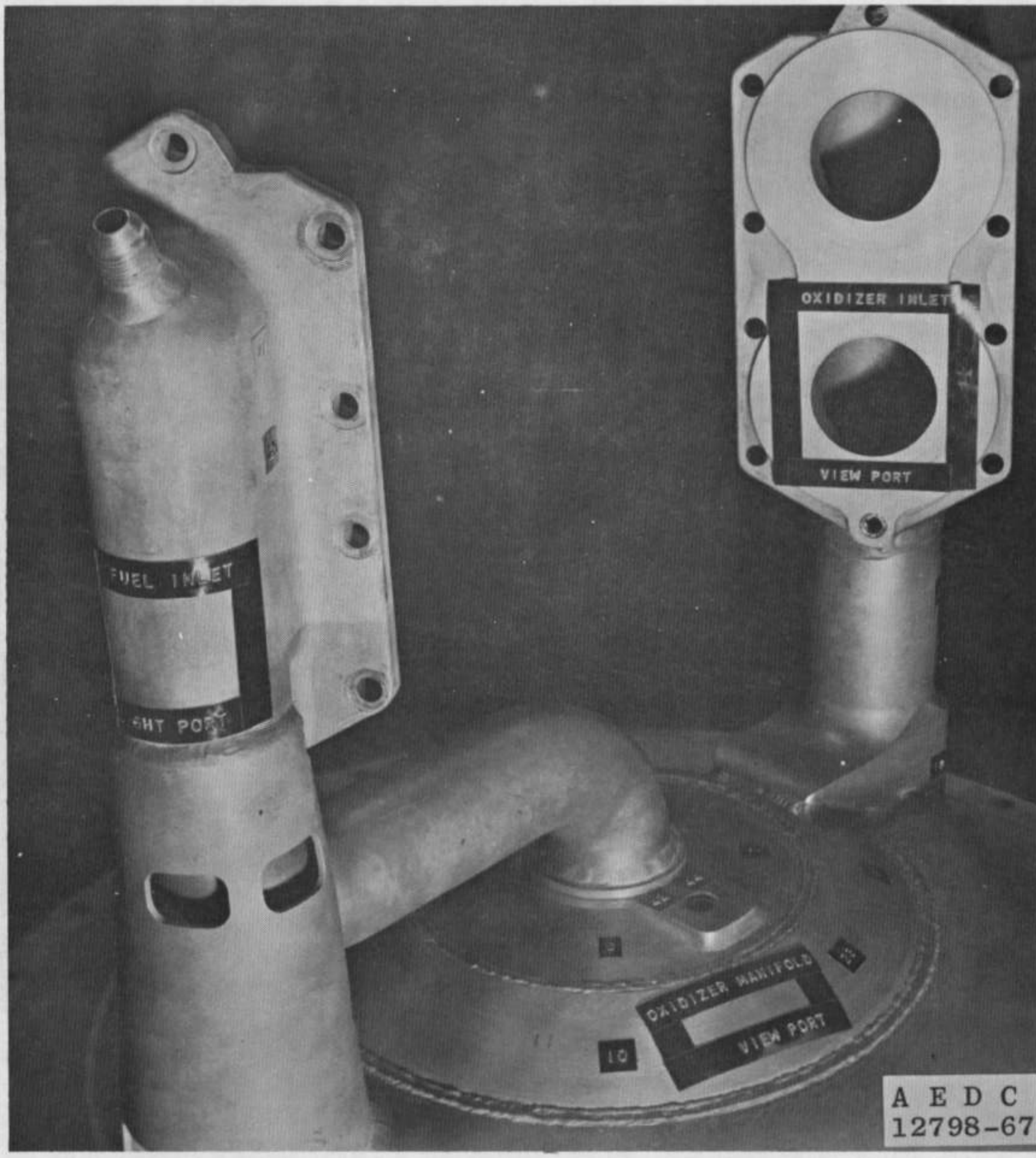


Fig. 3 View of SPS Injector Oxidizer Duct Showing Instrumentation and View Port Locations

Fig. 4 View of SPS Injector Face Showing Instrumentation and View Port Locations

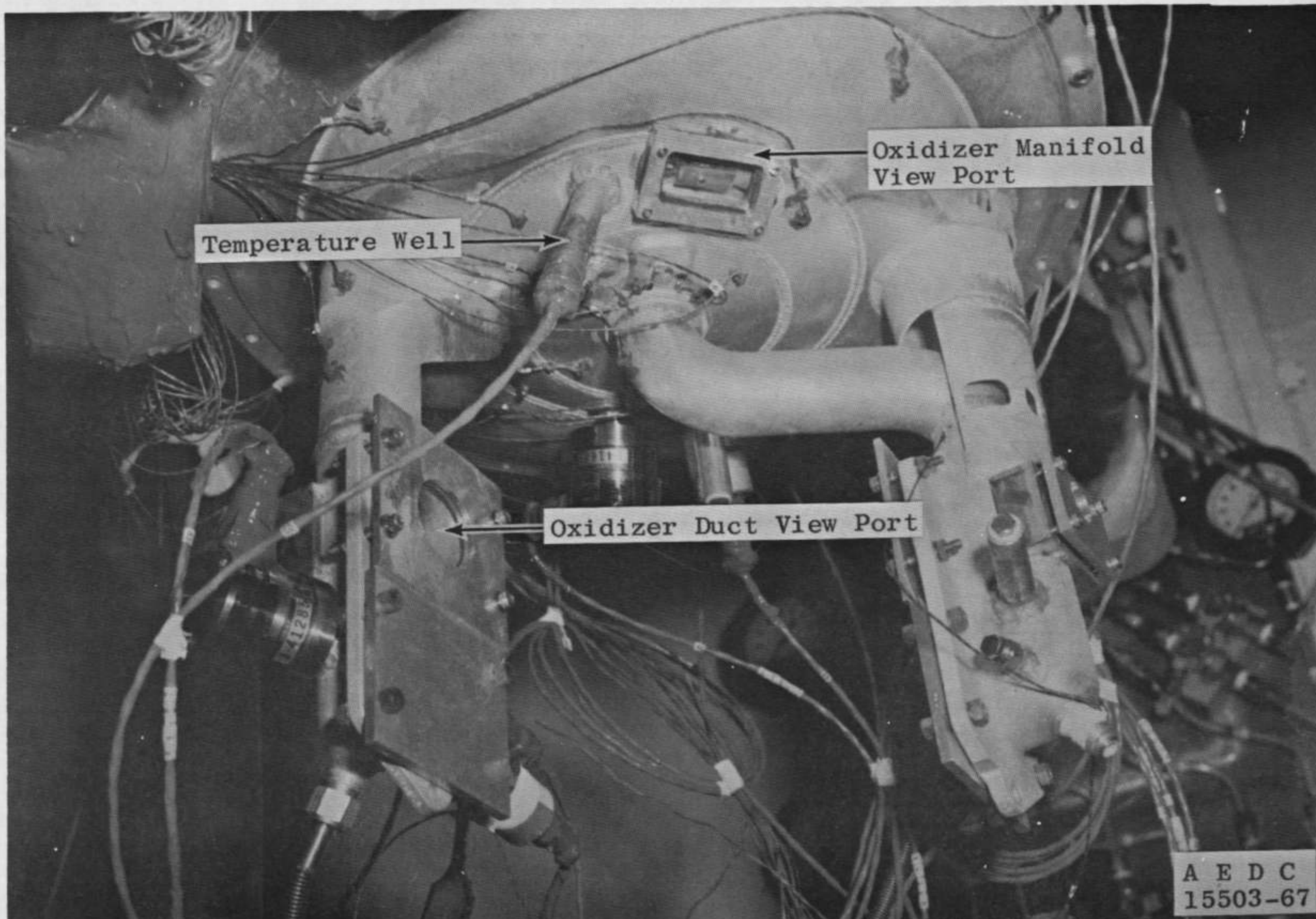


Fig. 5 Modified SPS Injector Showing Instrumentation and View Ports

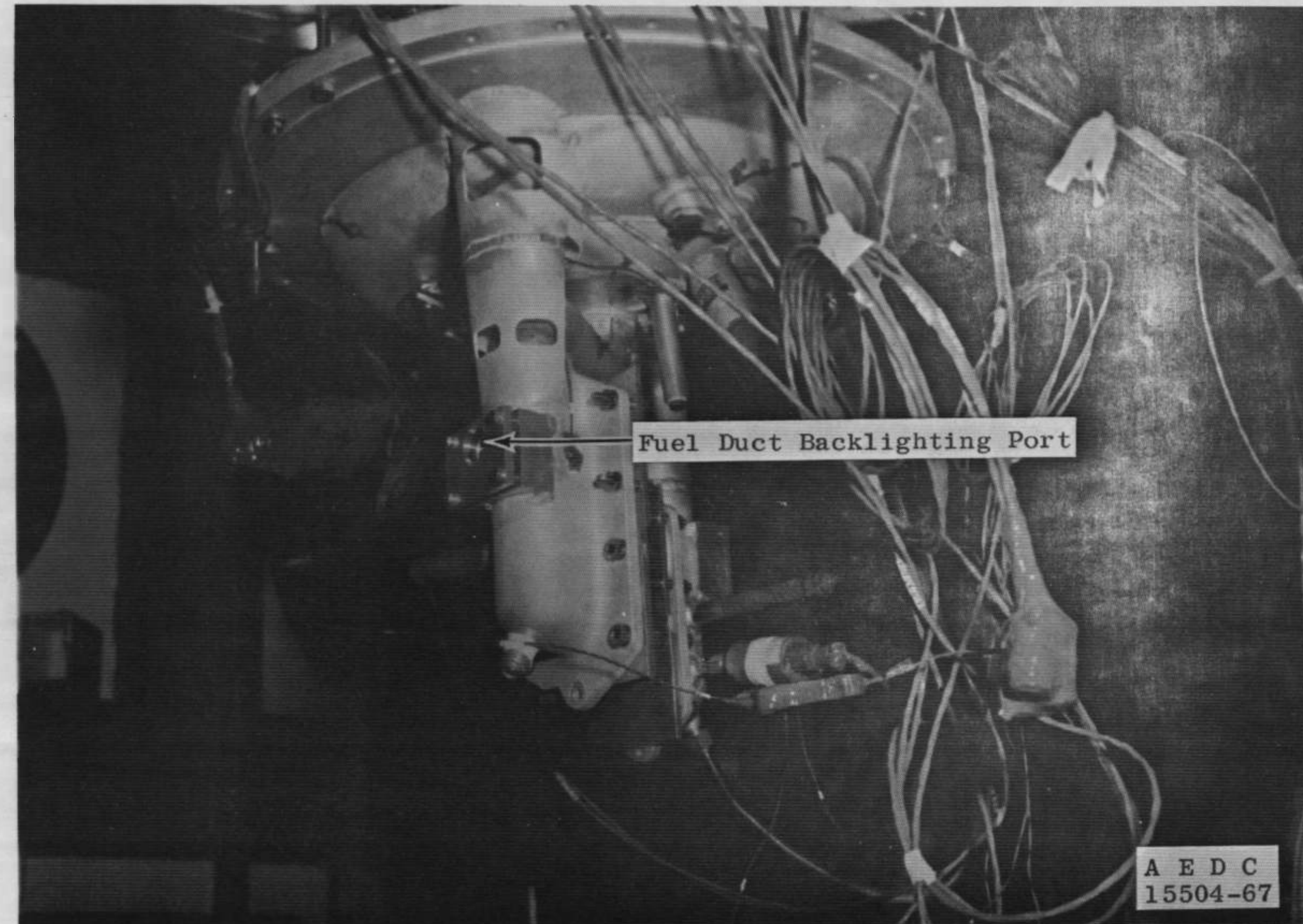


Fig. 6 Modified SPS Injector Showing Instrumentation and Backlighting Port

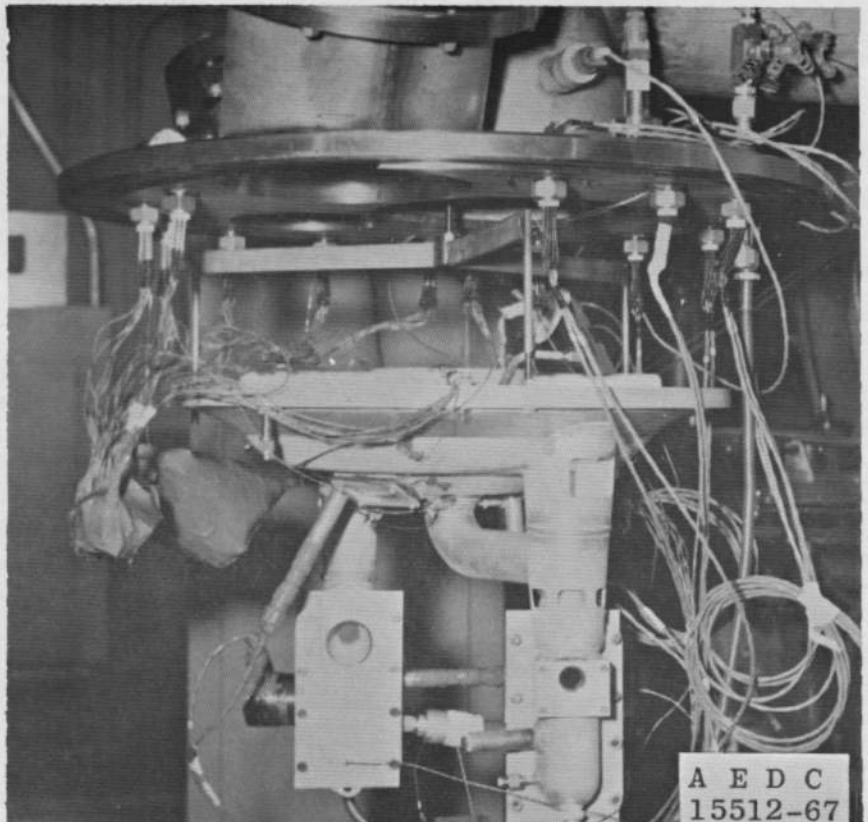


Fig. 7 SPS Injectar Shawing Instrumentation Attachments, Leads, and Feedthraughs with Antechamber Removed

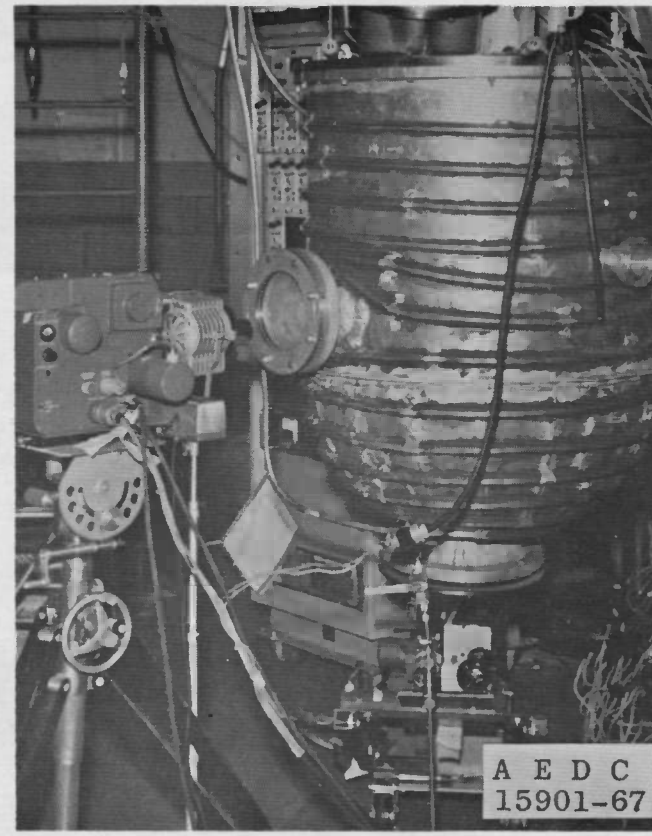


Fig. 8 Modified Antechamber after Installation - Ready far ARC (8V) Tests

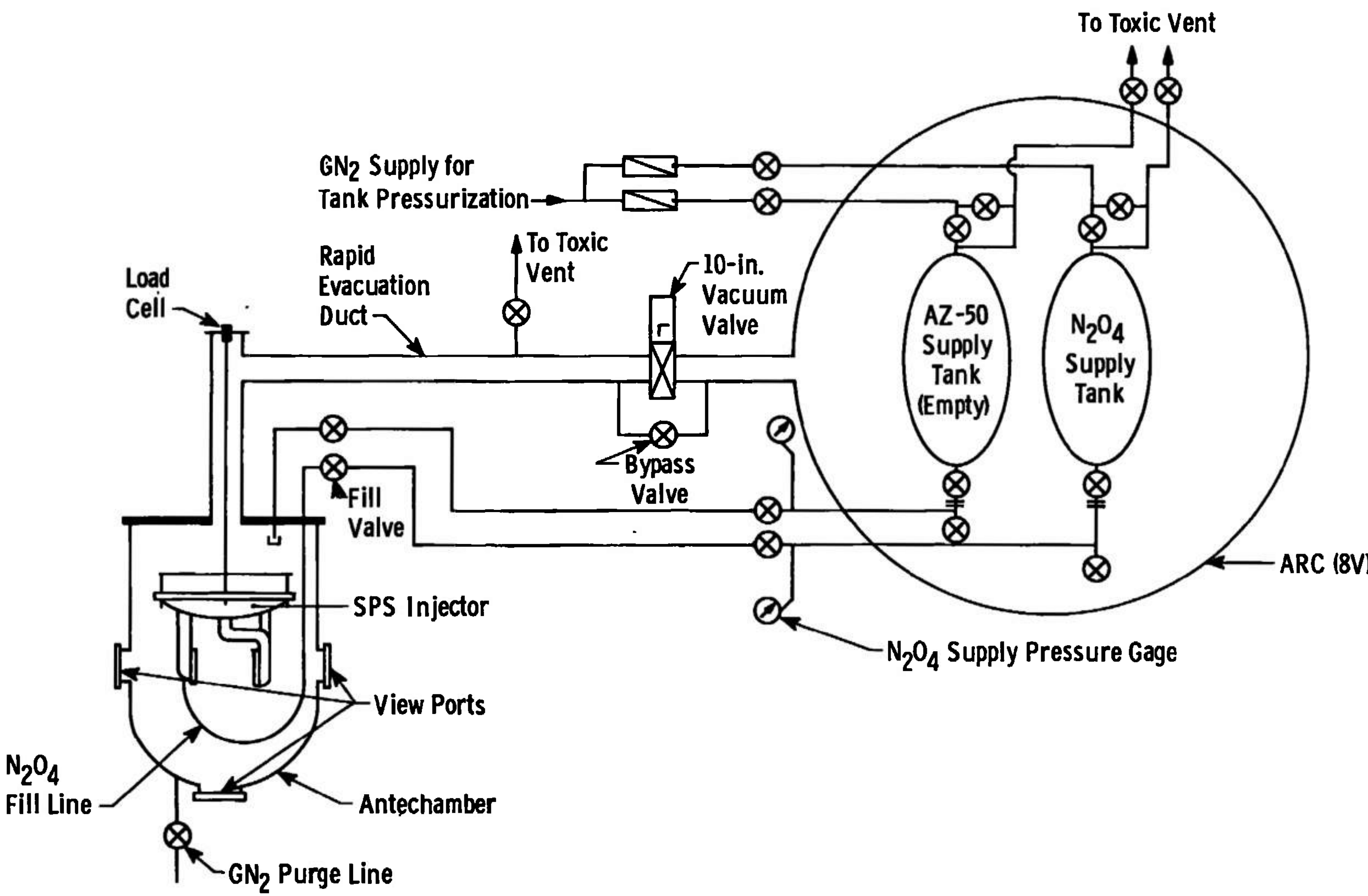


Fig. 9 Apollo SPS Injector Cold Flow (Phase I) Test Support System Diagram

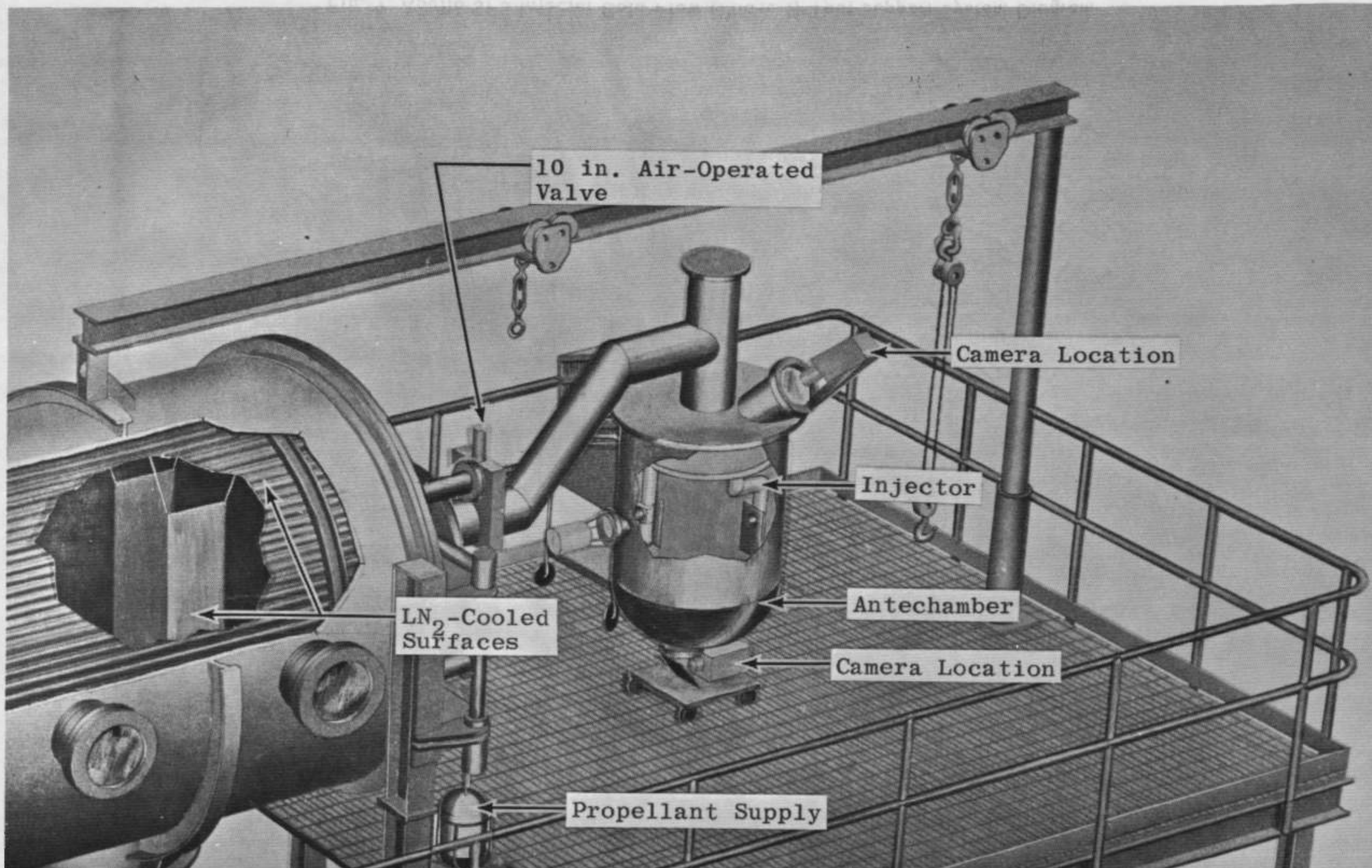


Fig. 10 Artist's Concept of ARC (7V) Test Installation (Phases II and III)

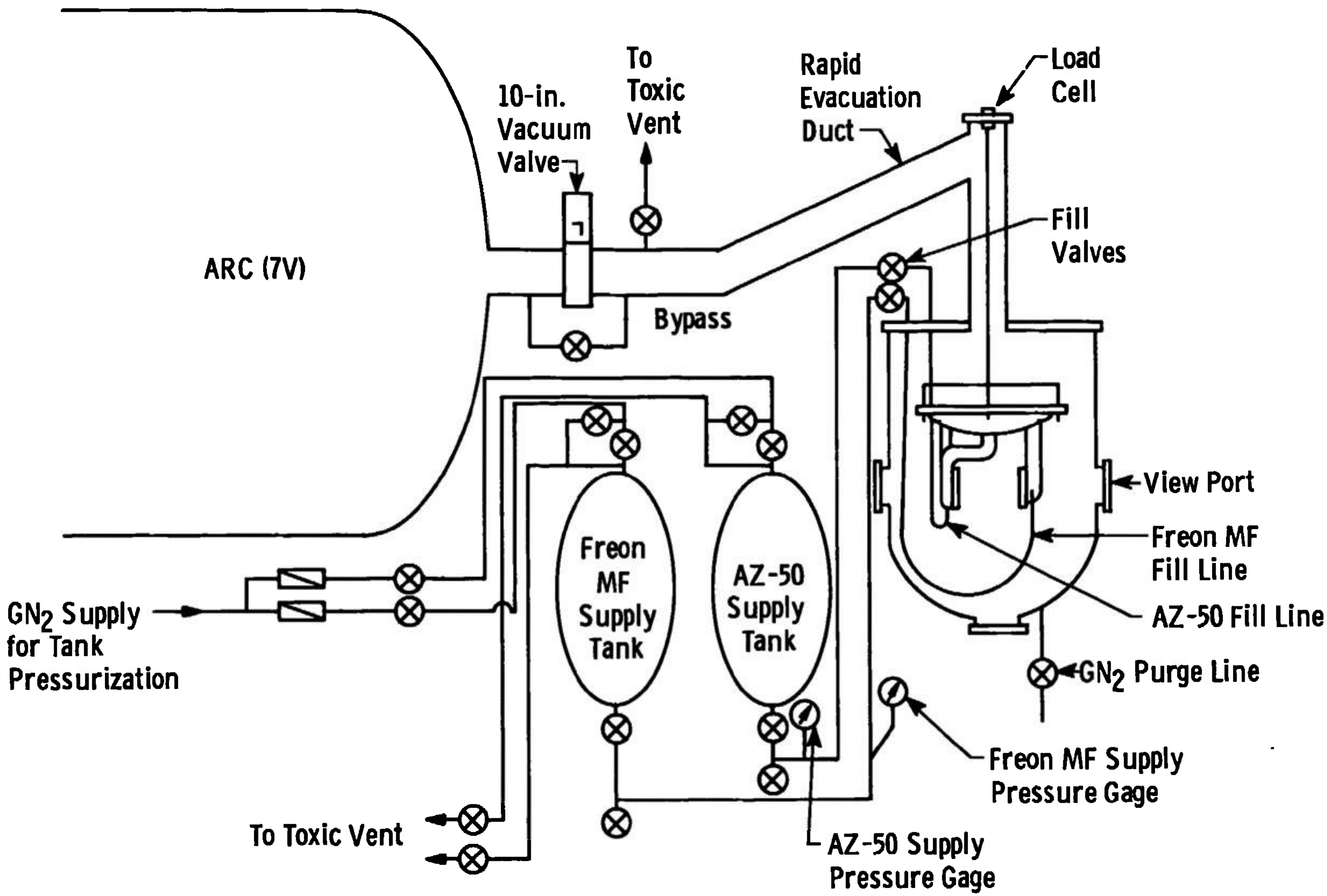


Fig. 11 Apollo SPS Injector Cold Flow (Phases II and III) Test Support System Diagram

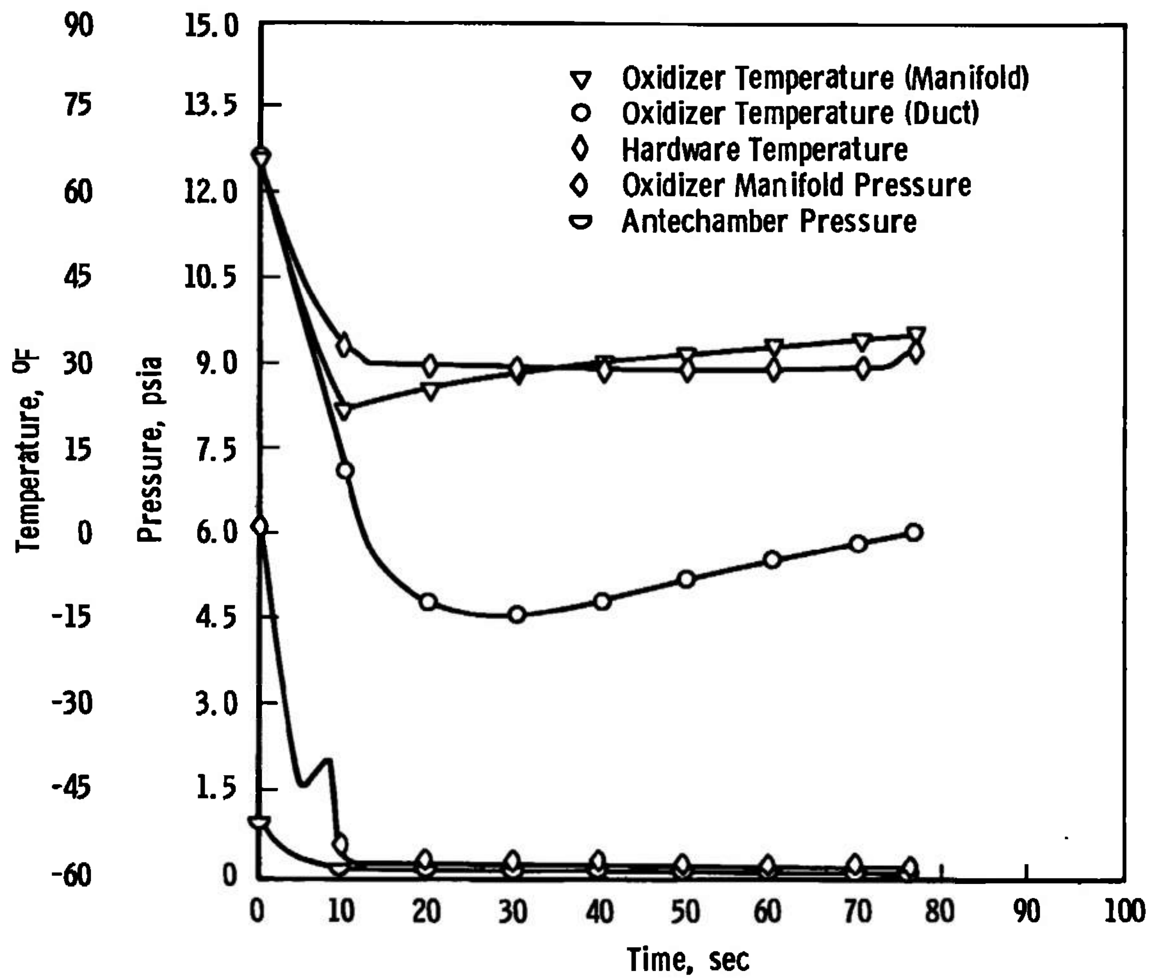


Fig. 12 Phase I, 62°F Oxidizer Test Run

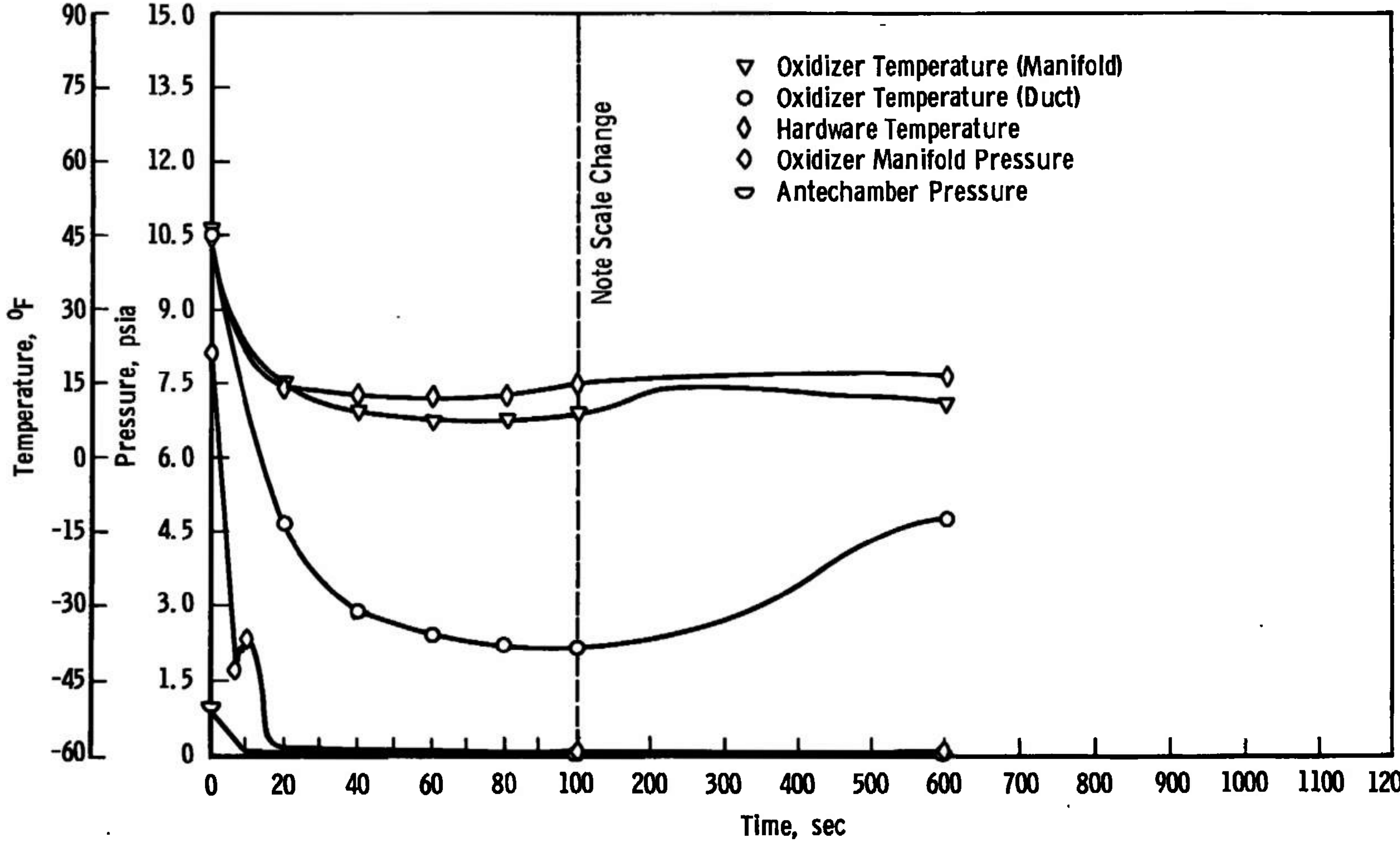


Fig. 13 Phase I, 45°F Oxidizer Test Run

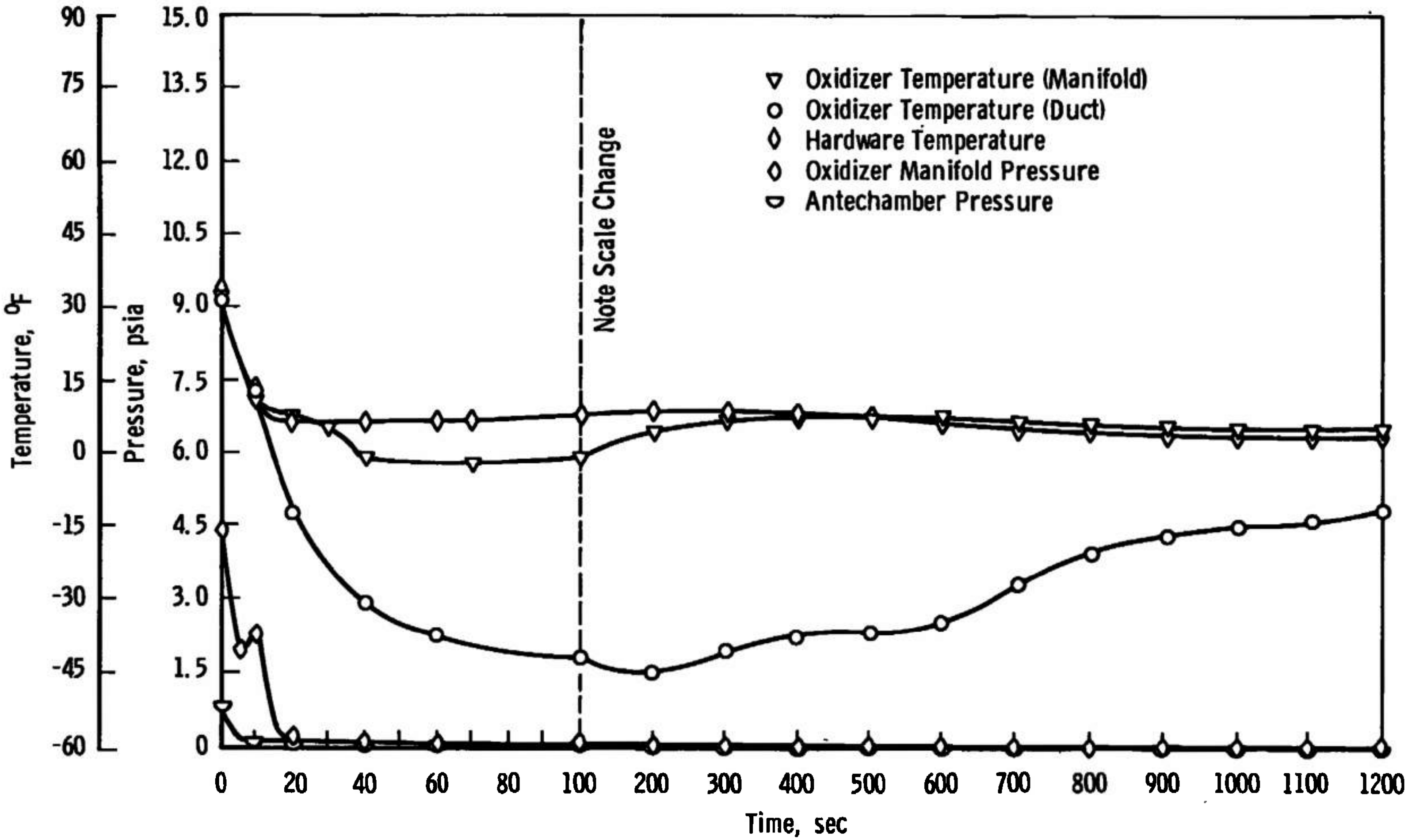


Fig. 14 Phase I, 33°F Oxidizer Test Run

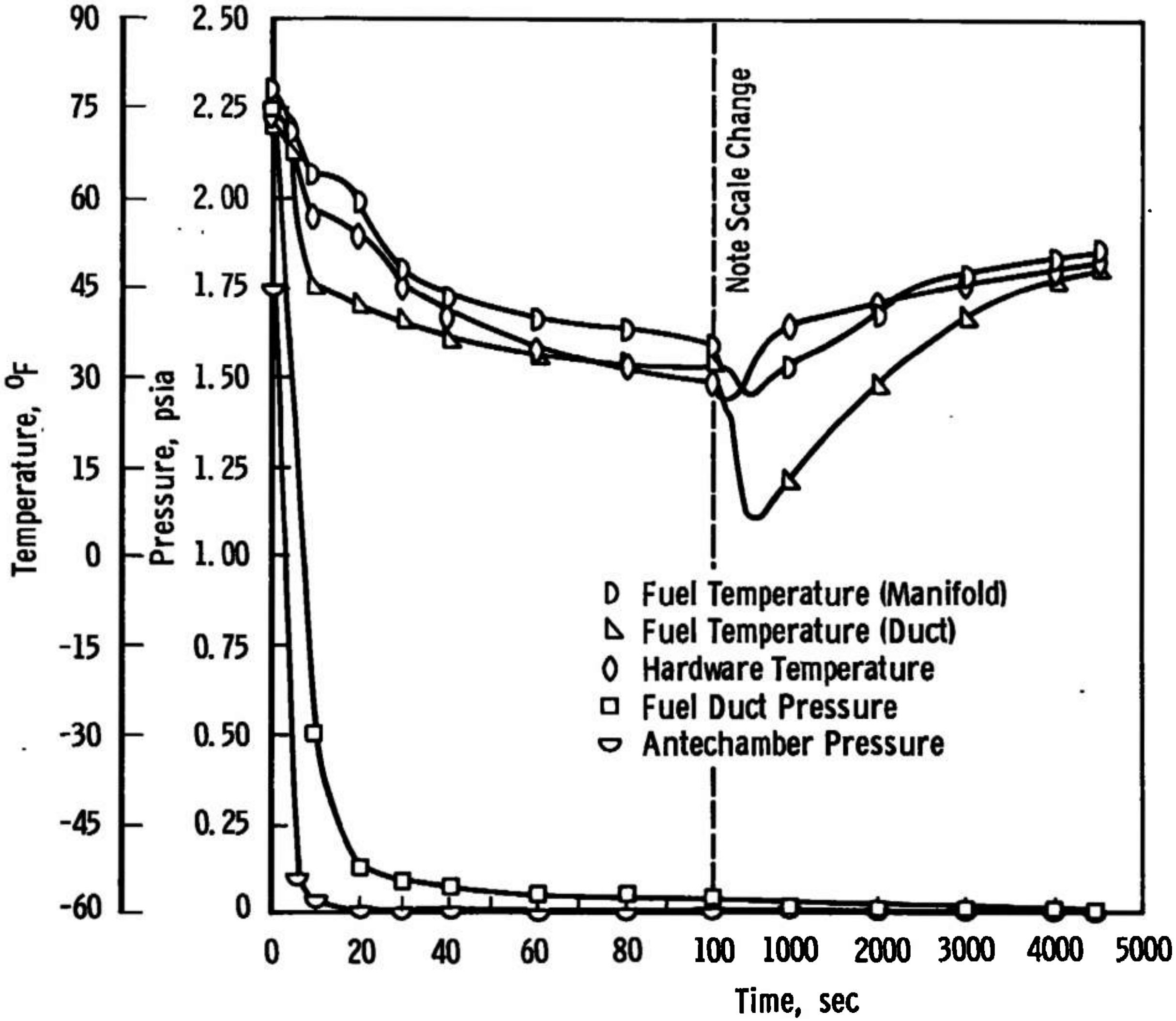


Fig. 15 Phase II, 75°F Fuel Test Run

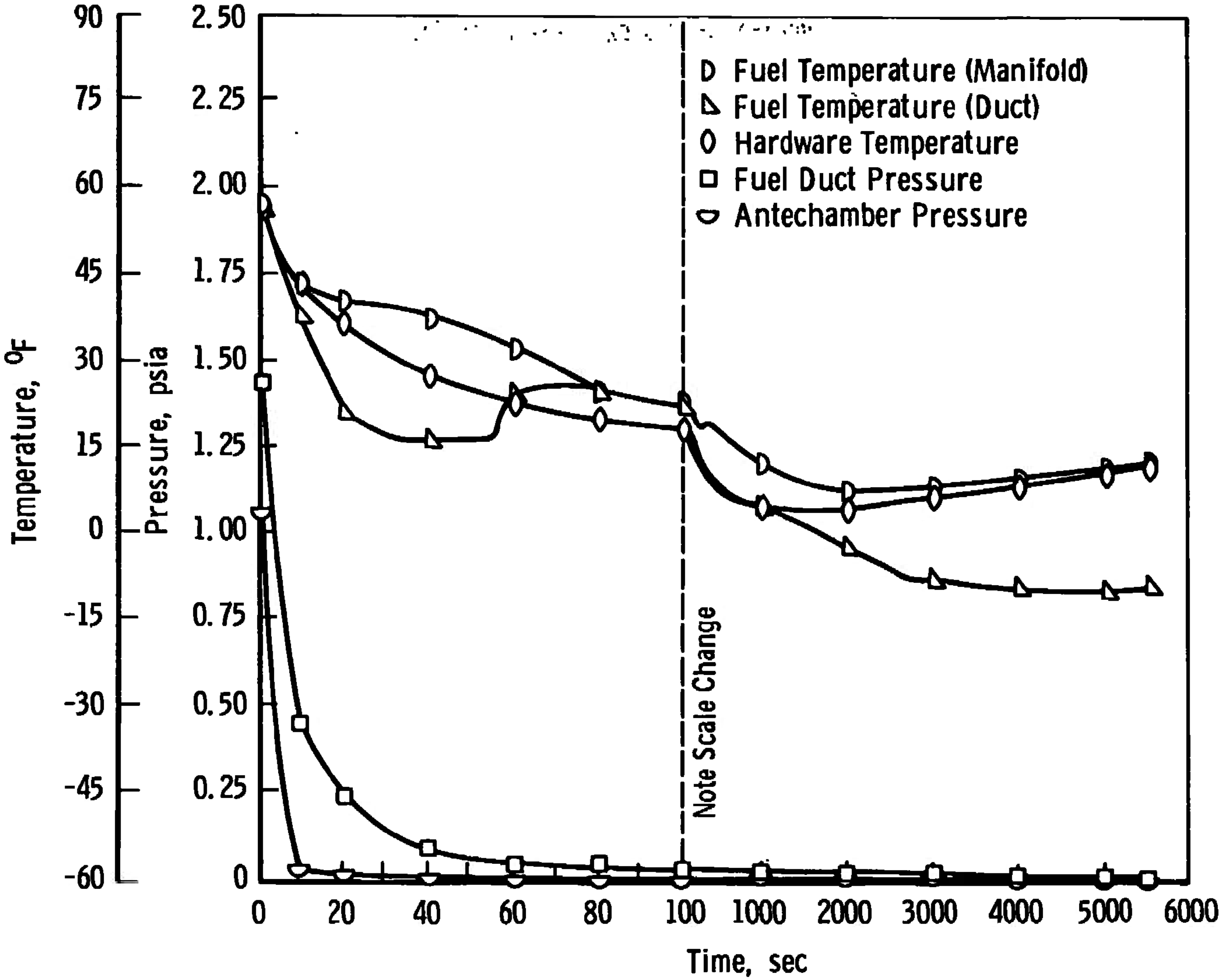


Fig. 16 Phase II, 52°F Fuel Test Run

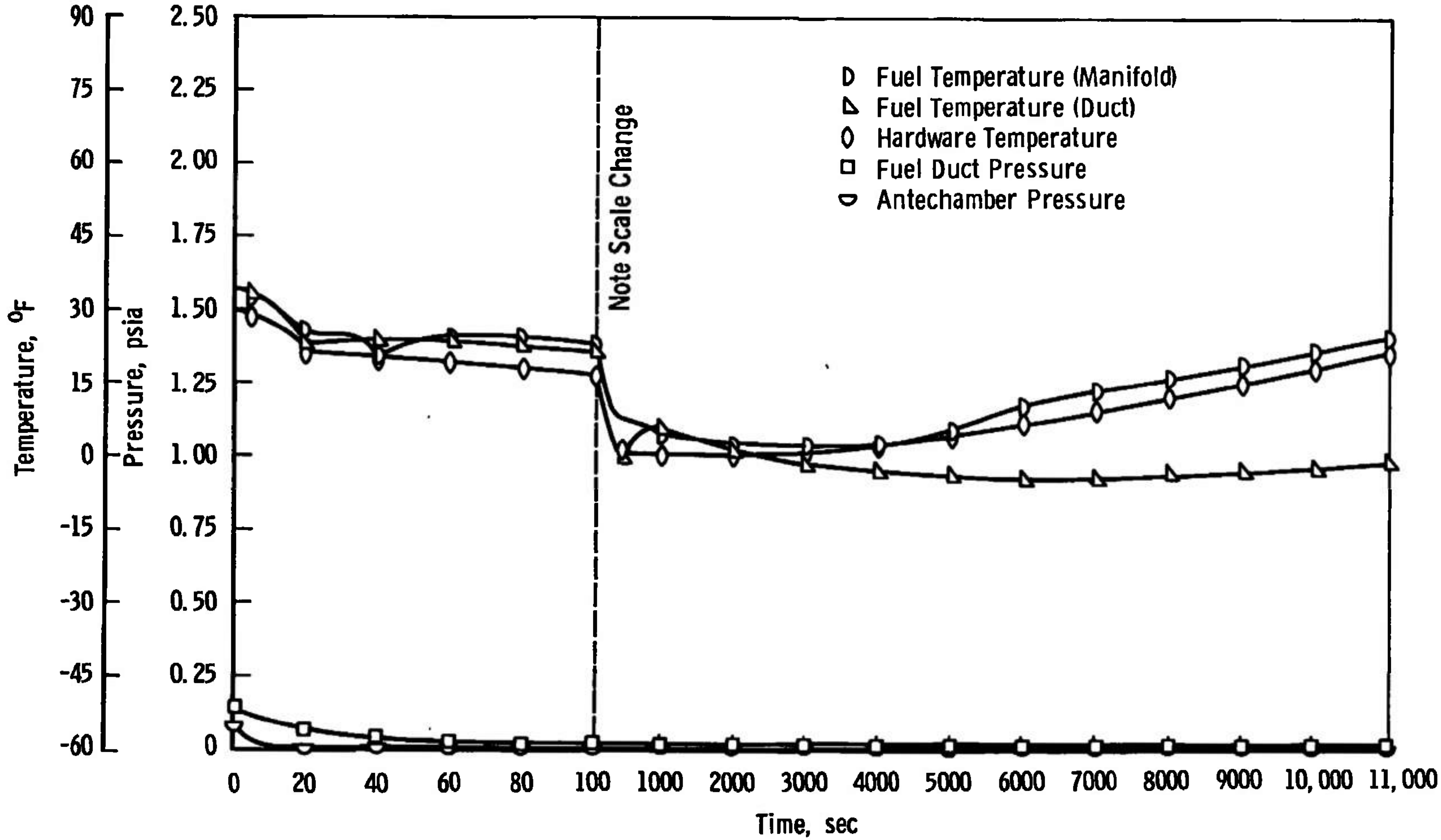


Fig. 17 Phase II, 33°F Fuel Test Run

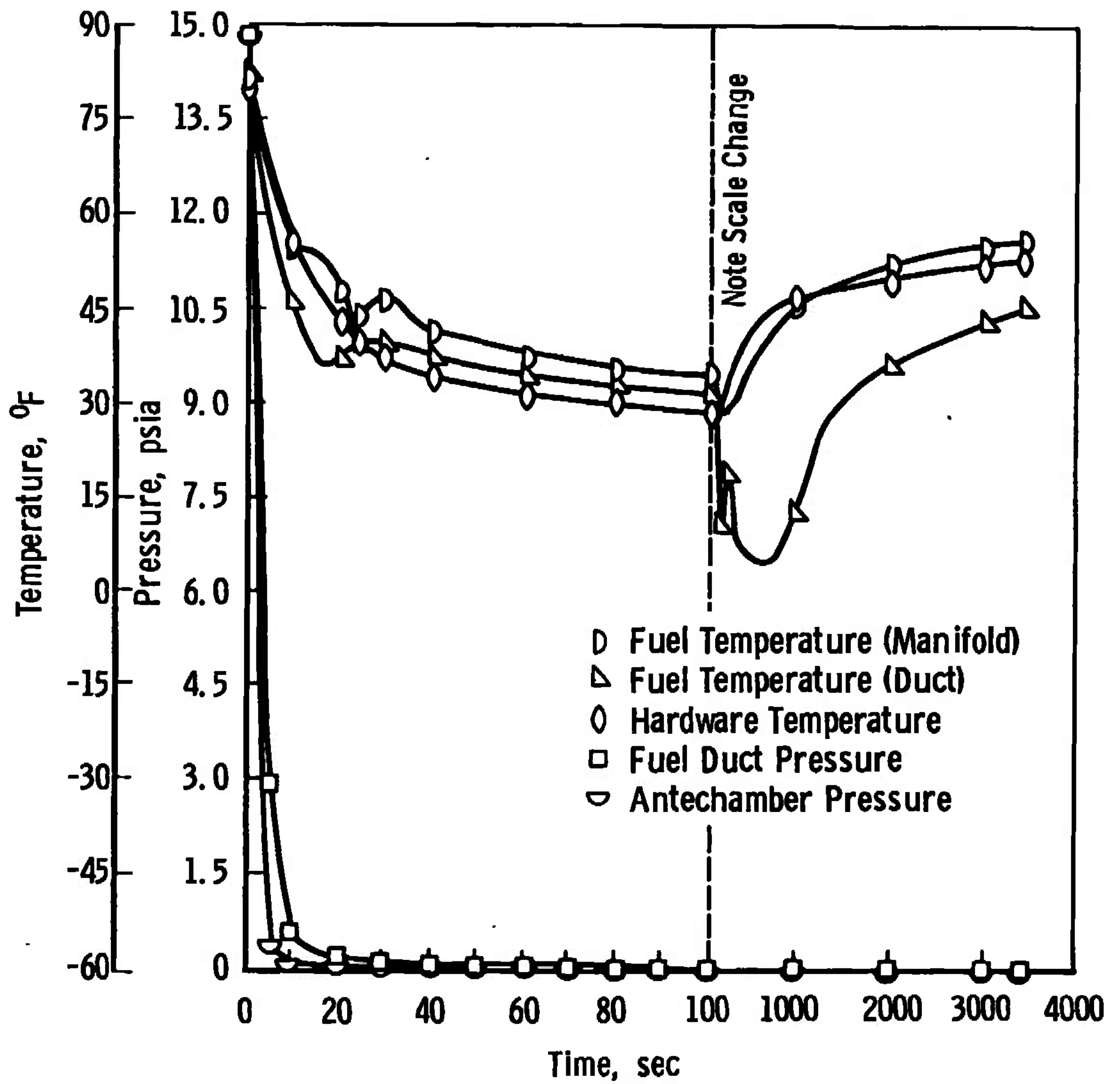


Fig. 18 Phase III, 78°F Fuel and Simulated Oxidizer Test Run

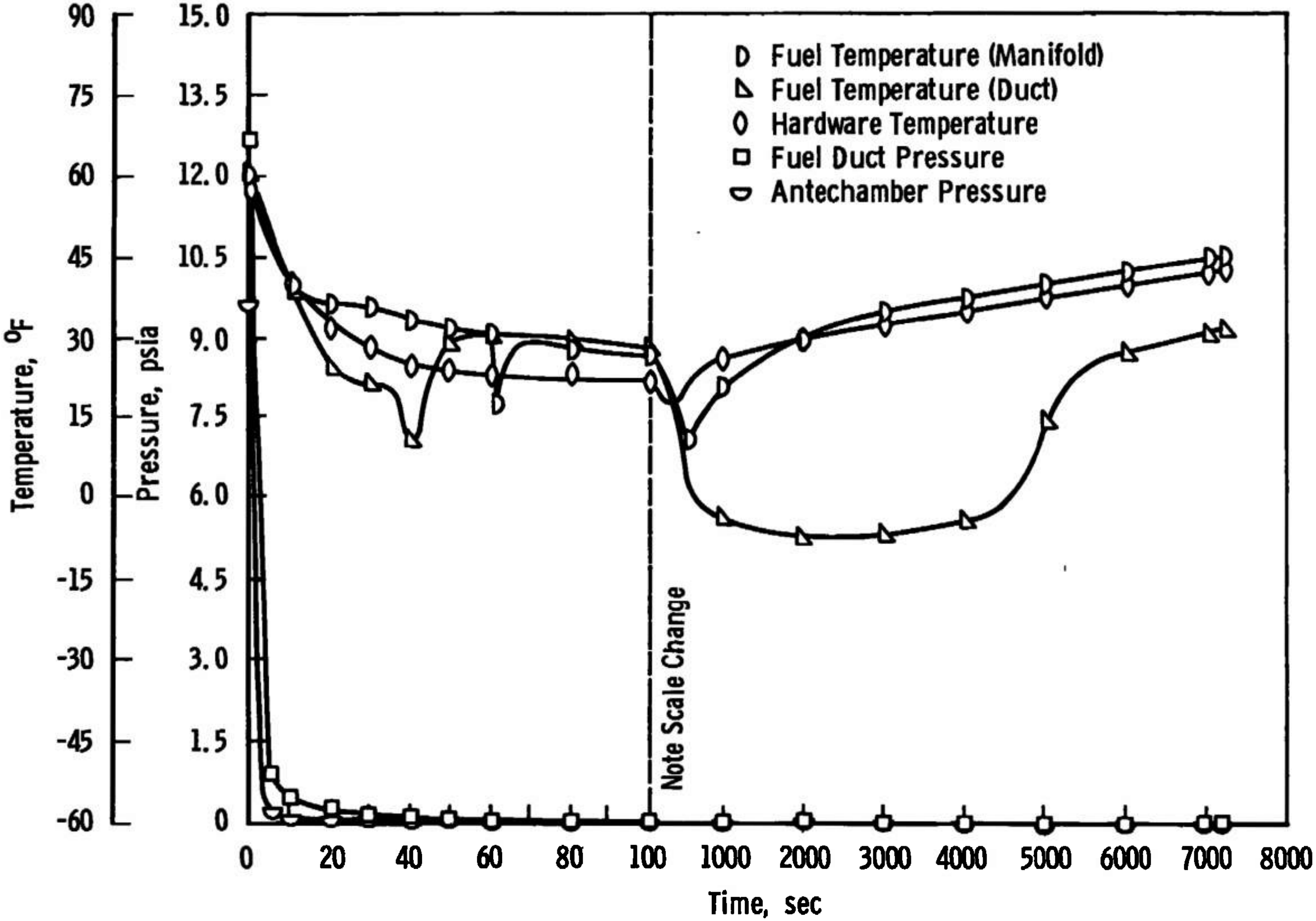


Fig. 19 Phase III, 56°F Fuel and Simulated Oxidizer Test Run

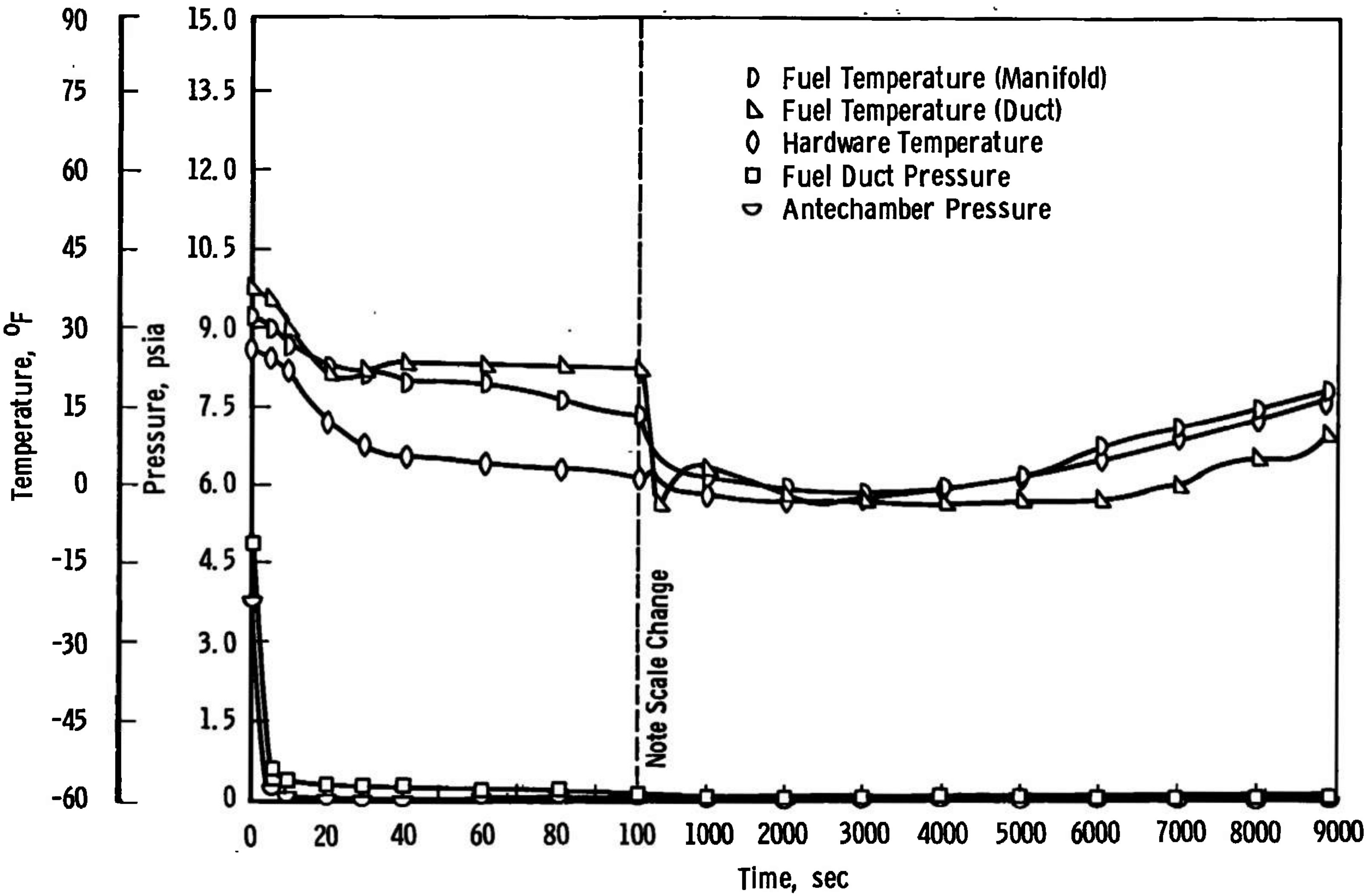
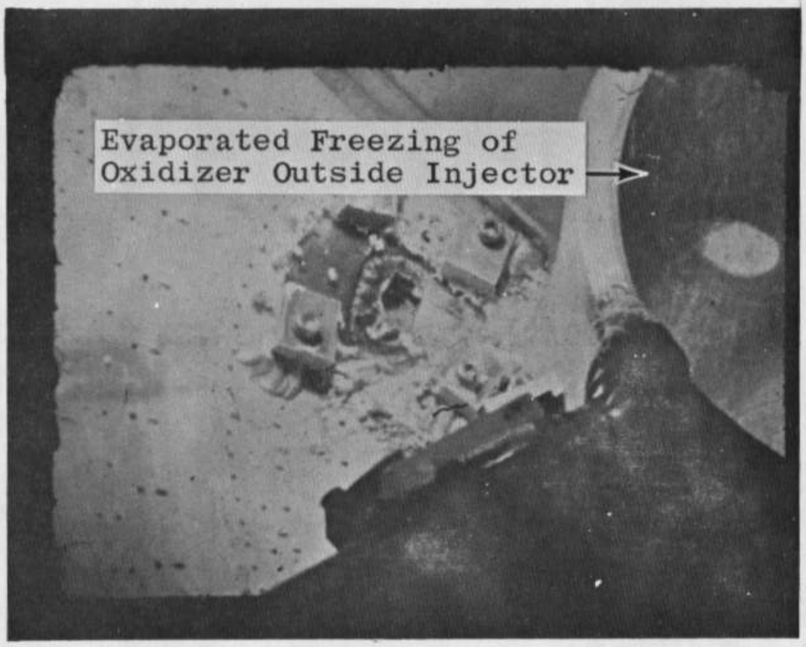
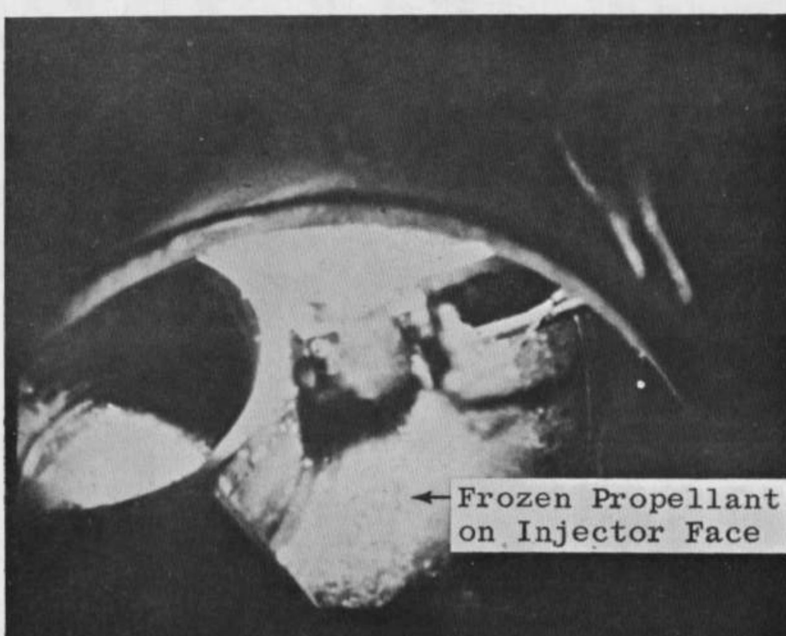


Fig. 20 Phase III, 25°F Fuel and Simulated Oxidizer Test Run

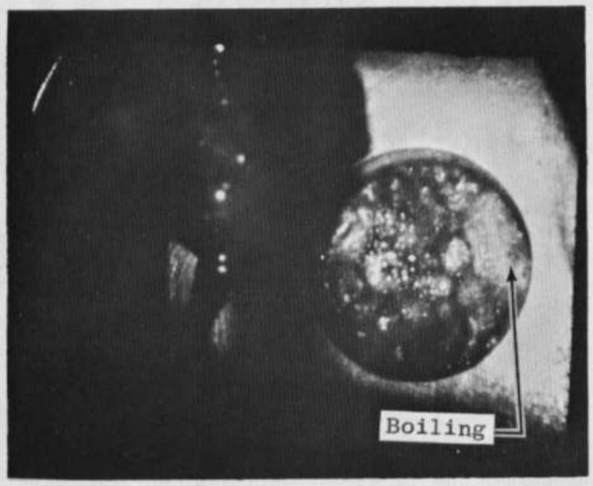


a. 45°F Oxidizer Test

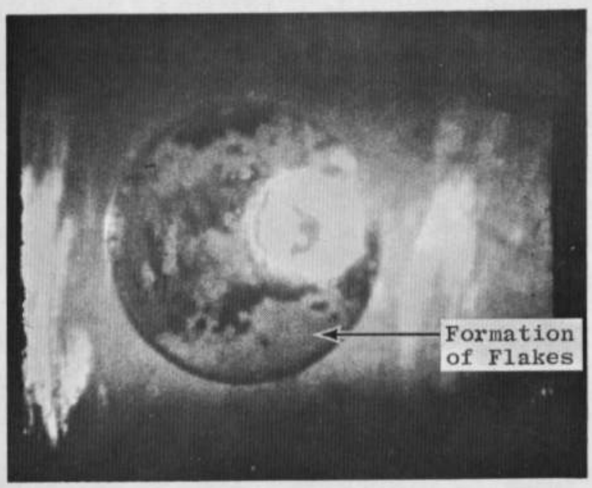


b. 30°F Combined Fuel and Simulated Oxidizer Test

Fig. 21 Top View of SPS Injector Showing Formation and Buildup of Ice



a. 55°F Fuel Test



b. 45°F Oxidizer Test

c. 30°F Fuel Test

Fig. 22 View of SPS Injector Fuel Duct at Various Conditions

TABLE I
APOLLO SPS INJECTOR COLD FLOW TEST INSTRUMENTATION SUMMARY

Temperature Measurements	Position Number	Type Sensor	Readout Channel Numbers
Injector Skin			
Fuel Duct	1, 2, 3	CATC①	UDS② 1, 20, 21
Fuel Cover	4, 5, 6, 7		M-1③/UDS 22, 23, 24, 25
Oxidizer Manifold (Inboard)	8, 9, 10, 11		UDS 26, 27, 30, 31
Oxidizer Manifold (Outboard)	12, 13, 14, 15		UDS 32, 47, 50, 51
Oxidizer Duct	17, 18, 19		UDS 53, 54, 55
Flange Edge	16		UDS 52
Liquid Temperature Probes			
Fuel Duct	20	PRTP④	12, 43, 74, 125
Fuel Manifold	21		13, 44, 75, 126
Oxidizer Duct	22		14, 45, 76, 127
Oxidizer Manifold	23		15, 46, 77, 130
Front Side of Injector			
Injector Face (Center Fuel Hole)	24	CATC	UDS 56
Injector Face (Center Oxidizer Hole)	25		57
Injector Face (Center Baffle Edge)	26		60
Injector Face (Radial Baffle Side)	27		61
Fuel Baffle (Feed into Transducer Fitting)	28	CATC Ceramo Probe	62
Fuel Channel (Most Inboard)	29	CATC	63
Fuel Channel	30		100
Fuel Channel	31		101
Fuel Channel (Most Outboard)	32		102
Oxidizer Channel (Inboard)	33		103
Oxidizer Channel (Outboard)	34		104
Antechamber, Inside Skin	35	CATC	M-2/UDS 105
Antechamber, Inside Skin	36		UDS 106
Antechamber, Inside Skin	37		UDS 107
Antechamber, Inside Skin	38		UDS 110
Antechamber, Inside Gas	39		M-3
Antechamber, Inside Gas	40		111

Pressure Measurements	Position Number	Type Sensor	Pressure Range, psia	Readout Channel Numbers
Fuel Injector Cavity				
Fuel Inlet (High Pressure)	41	Statham 15 psia	0-10	Oscillograph - 1
Fuel Manifold (High Pressure)	42		0-10	Oscillograph - 3/UDS 2, 33, 64, 115
Fuel Inlet (Low Pressure	43	CEC 1 psia	0-0.5	UDS 4, 35, 66, 117
Fuel Manifold (Low Pressure)	44		0-0.5	UDS 5, 36, 67, 120
Oxidizer Injector Cavity				
Oxidizer Inlet (High Pressure)	45	Taber 50 psia	0-30	Oscillograph - 5
Oxidizer Manifold (High Pressure)	46		0-30	Oscillograph - 7/UDS 3, 34, 65, 116
Oxidizer Inlet (Low Pressure)	47	CEC 1 psia	0-0.5	UDS 6, 37, 70, 121
Oxidizer Manifold (Low Pressure)	48		0-0.5	UDS 7, 40, 71, 122
Antechamber, Top Flange	49	Taber 100 psia	0-30	Oscillograph - 11/UDS 11, 42, 73, 124
Antechamber, Top Duct	50	CEC 1 psia	0-0.5	UDS 10, 41, 72, 123
Antechamber, Side	51	CEC 1 psia	0-0.5	Oscillograph - 9
Antechamber, Top Duct	52	CEC 0.5 psia	0-0.1	UDS 17

Force Measurements	Position Number	Type Sensor	Range	Readout Channel Numbers
Injector Weight	53	BLH/load cell 100 lb	0-70 lb	UDS 16

NOTES: ①Chromel®/Alumel® Thermocouple

②Universal Data System Readout Number

③Multipoint Readout Number (Redundant Readout in Chamber Control Area)

④Platinum Resistance Thermometer Probes

TABLE II
KEY MEASUREMENTS OF TEMPERATURE, PRESSURE, AND TIME

Temperatures, °F						Time to Lowest Temperature, sec			Initial Pressure, psia
Initial			Lowest						
Hardware	Manifold	Duct	Hardware	Manifold	Duct	Hardware	Manifold	Duct	
Oxidizer Tests									
62	65	66	29	21	-15	35	5	22	14.4
45	45	46	12	7.4	-38.6	50	75	97	8.4
33	31	33	6 and 5	-2.4	-45	60 and 1110	85	205	5.7
Fuel Tests									
75	78	76	25	26	5.5	234	519	594	2.25
54	56	55	3.7	6	15, -10	1620	2160	33, 4740	1.42
33	32	33	0	2	0, -4.5	2000	3400	420, 6800	0.62
Combined Fuel and Simulated Oxidizer Tests									
78	80	81	27	24	4.6	195	145	565	14.2
56	58	60	17.7	18.5, 11	10, -7	300	60, 540	41, 2460	12.8
25	30	36	-3	-1.70	18, -4	2100	3720	22, 340, 4440 T = -4°F	3.4

TABLE III
EVENTS AS RECORDED BY PHOTOGRAPHIC COVERAGE

Initial Hardware Temperature, °F	Time for Liquid Level to Drop below Duct View Port, sec	Time to End of Significant Ice Activity, sec
	Oxidizer Tests - Phase I	
62	3.2	12
45	4.5	11
33	2.5 - Slush Starting to Form	10 - Large Pieces of Ice Falling from Upper Areas - Buildup of Ice in Duct - Restart Questionable
	Fuel Tests - Phase II	
75	3.0	50
54	9.0	53
33	18.0 - Slush Starting to Form	34 - View Port Completely Blocked - No Restart
	Fuel and Simulated Oxidizer Tests - Phase III	
78	9.5	75
56	10.0	60
25	10 - Level Visible Near Bottom of View Port	20 - Ice and Slush Forming in Window and Continuing to Form through 150 sec of Film Coverage - No Restart

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13. ABSTRACT

A full-scale production Apollo Service Propulsion System Injector was modified to accommodate detailed instrumentation and visual observation capability during a series of propellant rapid expansions to high vacuum conditions to determine the venting characteristics of the injector. It had been suspected that after short burns of the engine at altitude, evaporative freezing of the residual propellants in the injector might result in clogged passages which could prevent safe restarts for extended time periods. Test results indicate that 5 min of venting between engine firing is adequate if propellant and injector temperatures are maintained above 55°F.

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14. KEY WORDS	LINK A		LINK B		LINK C	
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1. Missiles — Apollo
2. Injectors — Performance

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